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Lewis Research Center
Cleveland, Ohio

TECHNICAL PAPER proposed for presentation at
Fifth Electric Propulsion Meeting sponsored by the
American Institute of Aeronautics and Astronautics
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ABSTRACT

An ion thrustor system, including a new type of mercury feed system and a shielded neutralizer, has been designed and tested at a thrust level of 0.5 millipound. The use of a radial flow propellant distributor and an oxide-coated brush cathode resulted in discharge power losses one-fourth of those previously encountered at this thrust level. Different lengths and diameters of the discharge chamber were tested to establish a compromise between discharge power losses and propellant utilization. The final design discharge chamber was 5-centimeters in diameter and 7.5-centimeters long. The complete flight-type thrustor system used a permanent magnet field and weighed (without propellant) 1.36 kilograms. The power to thrust ratio, including power to heat the feed system and the neutralizer, was 222 watts per millipound at a thrust of 0.65 millipound and a specific impulse of 3050 seconds. The oxide-coated brush cathode was endurance tested 1553 hours in a thrustor and an identical cathode was heat cycled in a separate test for 418,000 cycles before failure.

INTRODUCTION

The primary objective of this work was to demonstrate mercury electron-bombardment ion thrustor performance suitable for station keeping and attitude control of satellites. For simplicity, the same basic thrustor design and thrust level were contemplated for both the station-keeping and attitude-control functions. No attempt was made to design a complete array of thrustors for a particular mission, but rather to build one unit which could be easily mounted in conjunction with other thrustors to provide a back-to-back station-keeping unit and/or an attitude-control thrustor.

A thrust level of 0.5 millipound was selected as being representative of a number of applications for a satellite weighing 500 to 1500 pounds. The duty cycle was estimated (ref. 1) at about 0.5 for a station-keeping function and about 0.01 for attitude control. A satellite lifetime of 3 years would thus require the operating lifetime of a station-keeping thrustor to be about 13,000 hours and include 1000 on-off cycles. A total of 50,000 on-off cycles was estimated for each of the attitude-control thrustors, with the corresponding lifetime of about 260 hours.

Previous operation of a mercury electron-bombardment ion thrustor has attempted to achieve the maximum thrust and high propellant utilization required for a primary thrust mission. The lowest thrust level obtained with efficient operation was about 1.5 millipounds (ref. 2). The present application required not only a reduction in thrust level without a severe loss of power or propellant efficiency, but also that the thrustor meet the lifetime and cycling requirements. Of prime concern were the erosion of the cathodes and (secondarily) the erosion of the accelerator grids. In a parallel research program, an oxide-coated refractory metal brush was found to have the best lifetimes (ref. 3) of any cathode tested. From unpublished accelerator erosion measurements, the required diameter of the accelerator grid (and discharge chamber) was determined to be in the range of 3 to 5 centimeters to obtain a lifetime of 13,000 hours at a thrust of

0.5 millipound and a propellant utilization efficiency of 80 to 50 percent.

In the course of the research program, various discharge chamber diameters and lengths were tested to determine minimum discharge chamber losses and maximum propellant utilizations. Initially the program was conducted with an easily varied electromagnetic field, and later effort was concentrated on the use of permanent magnets to reduce weight, power and system complexity. The mercury propellant feed system chosen was a positive-pressure liquid feed to a heated porous tungsten plug which controlled the vapor flow rate and separated the vapor-liquid phase. In addition, the porous plug served as an on-off valve because neglible flow passed through the plug when it was cold. Performance data are presented for two flight-type thrustors as well as for an extended test with a preliminary design.

APPARATUS AND PROCEDURE

Figures 1(a) and 1(b) are photographs of the flight-type thrustor with an electromagnetic field coil, and figure 1(c) is a photograph of a permanent magnet version. Figure 2 is a schematic view of a thrustor, indicating the relative locations of the discharge chamber, cathode, distributor, magnetic coil, and accelerator grids.

Flight-Type Thrustor

The nominal size (anode diameter) of the flight-type thrustor was determined to a large extend by the lifetime requirements of the accelerator grid. Details of the accelerator gride diameter selection may be found in appendix A. Briefly, the grid erosion is directly related to the square of beam current density and inversely to the propellant utilization. For the desired lifetime of 13,000 hours and thrust of 0.5 millipound, the minimum accelerator diameter was determined to be 3 to 5 centimeters at propellant utilization efficiencies of 80 to 50 percent. The final accelerator design was 5 centimeters in diameter. The accelerator and screen grids were both fabricated of a 0.16-centimeter-thich molybdenum sheet. Holes were drilled in both grids on a 0.635-centimeter equalateral triangular spacing. The screen grid and accelerator holes were 0.476 and 0.317 centimeters in diameter, respectively. The accelerator holes were made smaller to both increase the web material between holes (thus increasing the lifetime) and to somewhat decrease the loss of neutral propellant through the grid system. The screen-accelerator grid separation was held at 0.15±.01 centimeter by shielded aluminum oxide ball insulators.

By using the results of the variable geometry thrustor tests, the discharge chamber was designed with an anode diameter D_a of 5 centimeters and a length L_a of 7.5 centimeters. All sheet metal parts were made of nonmagnetic stainless steel. A magnetic coil produced a tapered field with magnitudes of 56 gauss at the distributor and 24 gauss at the screen. The permanent magnets produced a field with a range of near zero to a maximum of 105 gauss. Four rod magnets, 0.785 centimeter in diameter and 8.25 centimeters long were located between mild steel pole pieces. This configuration appears in a photograph in figure 1(a) and is also sketched in figure 3.

The chamber cathode was a tantalum brush (0.5 cm in diameter and 1.2 cm long) coated with Radio Mix No. 3 (57 percent $BaCO_3$, 42 percent $SrCO_3$, and 1 percent $CaCO_3$) and had a surface area of 1.8 square centimeters. The cathode was supported between two copper rods and was centrally located in front of and parallel to the plane of the distributor (fig. 3). The cathode was approximately 1 centimeter from the distributor. The cathode brush diameter was sized large enough to make the required lifetime feasible and yet small enough to reduce the thermal losses. The length was determined by the emitting area required. This surface area was actually larger than nominally required (nominal emission, 1 A/sq cm) because temperature gradiants in a unit of this small size greatly reduced the emission at the ends.

The propellant distributor was of a radial type (first reported in reference 4) to increase the cathode lifetime and at the same time produce a high thrustor total efficiency. The inner hole diameter of the distributor plate was 2.54 centimeters, and the distance between the cathode mounting block and this plate was about 0.32 centimeter. Other types of propellant distribution were not attempted in this program.

Propellant Feed System

In general, the feed system consisted of a spherical reservoir for liquid mercury (not shown in fig. 1(b)), a molybdenum tube leading to a heated porous tungsten plug, and a thermal isolation tube connecting the porous plug to the rear of the thrustor. Inside the thrustor and behind the distributor the mercury vapor passed through a fine screen which served to retard the flow of ions from the discharge chamber upstream into the higher pressure region near the porous plug (ref. 5).

The reservoir was a 7-centimeter-diameter hollow metal sphere. Part of the volume was charged with an inert gas to a pressure of 0.5 atmosphere to ensure positive flow to the porous plug. Gravity was normally used to maintain the position of the mercury in this sphere. A diaphragm would naturally be required for the zero-gravity environment of a space mission.

A single sphere of 7-centimeter diameter should have sufficient volume to hold both the inert gas and the mass of mercury required (about 1500 g) for one thrustor operating at 0.020-ampere beam current (0.5 mlb thrust at 4000 V net accelerating energy) for a 5000-hour test with 50-percent propellant utilization efficiency.

A molybdenum tube was chosen to duct the mercury from the reservoir to the porous tungsten plug both because of its resistance to corrosion by hot mercury and its compatibility withporous tungsten. A sketch of the tube, porous tungsten plug, and swaged nichrome heater is included in the RESULTS AND DISCUSSION section of this paper. The porous tungsten plug was 0.3 centimeter in diameter, 0.07 centimeter thick and electron-beam welded to the molybdenum tube. A thermocouple was spot welded on the side of the tip to indicate the operating temperature. Thermocouples were also spot welded along the molybdenum tube to measure thermal conduction. Porous tungsten materials with pore radii of 2 to 4 microns and densities of 70 to 80 percent were used to fabricate the porous plugs for the several vaporizers used. The size of the porous plug, molybdenum tube, and heater element was kept small to shorten the thermal time response and to reduce the operating power.

The porous plug discharged into the .025-centimeter wall thickness thermal isolation tube, which was 1.9 centimeters in diameter and 7.6 centimeters long. An additional swaged nichrome heater was placed at the middle of the thermal isolation tube. This heater served to further thermally isolate the thrustor and the feed system. With 3 watts added at the center of the tube, the thrustor end of the tube could vary from

300° K (cold start) to 530° K (equilibrium operating temperature), and the variation of the porous plug temperature for a constant plug heater input would only be 550° to 553° K. This small rise in plug temperature represented only a 10-percent increase in the propellant flow rate. If no changes occur in the flow due to possible plug conductance or emissivity changes of the vaporizer surfaces, only an open loop control would be required to maintain the propellant flow within this 10-percent variation. For performance or endurance tests the actual porous plug temperature was used together with a calibration curve to determine the propellant flow rate.

Neutralizer

The neutralizer cathode consisted of a tantalum brush 0.25 centimeter in diameter and 0.7 centimeter long, coated with Radio Mix No. 3 oxides. As the emission density of the neutralizer cathode was much less than that of the discharge chamber, the important design consideration was not emission limitations but rather cathode reliability and space-charge limitations. The neutralizer was heated by passing an alternating current through the core wires and was supported by two copper rods. Two neutralizers (shown in fig. 1(a)) were used for redundancy. A heavy boron nitride block was placed between the neutralizer and the accelerator grid to prevent direct beam impingement on the neutralizer. The boron nitride block protruded 0.1 centimeter more than the edge of the cathode into the beam. The neutralizer cathode was located approximately tangentially to the nominal (2.5-cm. radius) edge of the beam and 2.8 centimeters downstream of the accelerator grid. A floating target (collector) was placed approximately 1 meter downstream of the thrustor in the path of the ion beam and its potential was monitored by a dc voltmeter. Since the neutralizer cathode was always grounded, the voltage of the target approximately represented the coupling voltage between the neutralizer and the ion-beam-plasma potential (which was within a few volts of the floating target).

Variable Geometry Thrustors

Several different variable geometry thrustors were used during the initial phases of the program, but they all may be schematically represented by figure 2. A steam-heated mercury vaporizer was used to supply propellant through a calibrated orifice. Often no neutralizer was used, and neutralization was obtained by grounding the collector or beam target. Large magnetic coils were interchanged to produce the proper fields with the different size discharge chambers that were tested. Through the use of matched flanges a single set of screen-accelerator grids were used with both the 5- and the 2.5-centimeter-diameter anodes. Accelerator grids 0.16, 0.32, and 0.64 centimeter thick were also tested. All the screen and accelerator grids had the same equilateral hole spacing and hole diameters as the flight-type thrustor. The chamber cathode was a thick-oxide-layer type (ref. 4) composed of a tantalum heater ribbon wrapped with tungsten wire and coated with Radio Mix No. 3. The cathode was shaped in a loop as shown in figure 2 with a total length of 2.5 centimeters and a surface area of 1.3 square centimeters. The thick-oxide-layer cathode was used because brush-type cathodes were not available for these early tests.

Vacuum Facility

Most of the tests with either the variable geometry or flight-type thrustors were conducted in a 0.5-meter-diameter bell jar connected to a large vacuum tank. Some of the neutralizer tests were performed with the thrustor completely inside the vacuum tank. The vacuum tank was 1.5 meters in diameter and 4.5 meters long. It was pumped by four 0.8-meter-diameter oil diffusion pumps with liquid-nitrogen-cooled baffles. With a thrustor operating, the tank pressure was 1.0 to 6.0×10^{-6} torr, and the bell jar pressure was 1.0 to 2.0×10^{-5} torr.

RESULTS AND DISCUSSION

Discharge Chamber Diameter

Previous programs (refs. 2 and 6) had produced considerable data with thrustors of 5, 10, and 20 centimeters in diameter and beam currents of 0.030 to 0.125, 0.060 to 0.600 and 0.125 to 1.100 amperes, respectively. Continuation of this trend would yield a thrustor size of about 2.5-centimeter diameter to produce a 0.020-ampere beam. As a thrustor is scaled down, however, two losses begin to increase. One loss is from an increased ratio of ion recombination at the walls because of a larger wall-surface-to-accelerator area ratio. The other loss arises from the use of a shorter discharge chamber length. Reference 7 indicates that reducing the chamber length below a certain value should adversely affect both the discharge power requirements and the propellant utilization. If the thrustor diameter is made larger (without increasing the beam current), the discharge plasma becomes more dilute, and high propellant utilizations are difficult to obtain at the lower ion-beam current densities (see eq. 34 or ref. 7). Requirements of accelerator grid erosion (appendix A) further restrict the selection of thrustor diameter to approximately 3 to 5 centimeters. Final selection of an anode diameter is thus a compromise between wall losses, plasma density, and accelerator grid erosion.

Discharge chambers with anode diameters of 2.5 and 5 centimeters were constructed. These two anode diameters were tested using the same accelerator grids, distributor plate, and cathode type and location. The result of these tests are shown in figure 4. The discharge energy per beam ion is plotted against the propellant utilization efficiency for discharge chamber diameters of 2.5 and 5.0 centimeters. The smaller diameter chamber had larger discharge losses at all propellant utilizations tested, and perhaps more important, it did not exceed a propellant utilization efficiency of 50 percent. The larger diameter chamber did not exceed a propellant utilization efficiency of 80 percent and the discharge chamber losses usually began to rapidly rise at values of 60 to 70 percent. The required strength of the magnetic field, as anticipated (ref. 6), was larger for the smaller diameter chamber. As shown in the next section, the effect of using different L_a/D_a values of 2.0 and 1.5 for this comparison was minor.

Discharge Chamber Length

Preliminary tests with a 5-centimeter-diameter thrustor (L_a/D_a equal to 1.0) indicated that the discharge losses (per beam ion) increased and the propellant utilization decreased as the thrust level was reduced to 0.5 millipound. A series of tests was therefore performed to measure the effect of an increased discharge chamber length on the performance of the thrustor.

Figure 5 is a plot of the discharge power per beam ion for various length discharge chambers with a 5-centimeter-diameter anode. The best discharge chamber performance was realized at L_a/D_a values of about 1.0 to

2.0. At higher values of L_a/D_a, the discharge losses rapidly increased. The maximum propellant utilization (not shown on fig. 5) was essentially unchanged by changes in the discharge length; at fixed propellant utilization, however, the discharge losses increased with increased length. The three curves presented in figure 5 each represent a different value of beam current (0.016, 0.023, and 0.030 A) and, hence, a different propellant utilization. The differences in the discharge voltages were caused by using this parameter to adjust the beam current. The differences in magnetic field intensity resulted from different coils being used for the different length chambers and are not considered significant.

On the basis of the data shown in figures 4 and 5, the flight-type thrustor was designed with a 5-centimeter diameter and an L_a/D_a of 1.5. This optimum value was close to the initial design of the 5-centimeter-diameter thrustor of reference 1 or 5. The lower values of discharge energies obtained in this investigation as compared with those of reference 1 or 5 were believed attributable primarily to the use of the radial flow distributor and partly to a larger (oxide-coated) cathode surface. However, the operation of a discharge chamber in even the best 0.5-millipound thrustor as compared to a 3-millipound or higher thrustor was poor due to unavoidable losses in propellant utilization and discharge power. For an electron-bombardment ion thrustor of the type reported herein, thrust levels below 10^{-1} millipounds appear to be impractical. Smaller chambers result in high wall recombination, and larger chambers (at this low thrust level) decrease the plasma density.

Accelerator Grid Thickness

There are two possible advantages in using a thicker accelerator grid. First, the lifetime is increased by adding more material to the grid. Second, the loss os neutral propellant from the discharge chamber would be expected to be somewhat decreased.

Accelerator grids, 0.16, 0.32, and 0.64 centimeter thick, were tested, and the accelerator impingement current of the three grids as a function of the net acceleration voltage is presented in figure 6. The normal thickness, 0.16 centimeter, produced the lowest impingement (about 1 percent of the beam current). The thicker grids produced successively higher impingement currents. The sharp rise with decreasing voltage at 2000 to 3000 volts was expected and was caused by local beam divergence due to space charge limits between the grids. The steep rise above 4 kilowatts for the thickest grid was probably the result of ions striking the downstream edges of the holes.

The thickness of the grids had negligible effect on the propellant utilization or the discharge chamber losses. The normal (and thinnest) accelerator grid was therefore used in the flight-type thrustor. At the measured impingement values, the estimated lifetime of the nominal grid was greater than the required 13,000 hours. Based on the results of figure 6 (where the minimum impingement increased strongly with thickness), no increase in lifetime would be expected for the thicker grids. With an increased test life, however, the downstream edges of the grid hole may erode away and the initially high impingement may drop to a lower value. Such tests as beveling the downstream edge of the accelerator grid holes were considered unnecessary in this investigation, although such modifications may result in improved (greater than 13,000 hr) accelerator grid lifetimes.

Flight-Type Thrustor

Discharge chamber. - Performance of the discharge chamber may vary depending on the condition or activation state of the oxide cathode. Both the thick-oxide-layer-type cathode, and the brush-type cathode used in this program possessed variations in emission characteristics with cathode condition. In general, the discharge chamber with a brush cathode performed better than an identically shaped discharge chamber with a thick-oxide-layer cathode. The thick-oxide-layer cathode was therefore not tested in the flight-type thrustor.

Figure 7 presents typical data for the brush cathode. Data are shown for four different values of ion-chamber discharge voltage and variable magnetic-field strengths at fixed values of propellant flow, accelerating voltages, and cathode heating power. The propellant flow rate of 0.038 ampere was defined as if each neutral mercury atom contained a single charge. There was a rapid dropoff in beam current and increase in discharge power per beam ion at values of magnetic field strength below 17 gauss. Above 17 gauss, there was an operating region where the discharge was relatively insensitive to magnetic field strength. The curves shown in figure 7 also indicate a higher discharge power per beam ion at higher values of ion-chamber discharge voltage. This may be somewhat misleading as the propellant utilization (beam current) was also increased, and higher discharge losses would be expected. A later figure will more clearly show the effect of propellant utilization on the discharge losses for the flight thrustor.

As a compromise between discharge losses and good propellant utilization, a value of beam current of 0.022 ampere, ion-chamber discharge voltage of 20 volts, and magnetic field strength of 20 to 30 gauss could be chosen as an optimum. The value of 900 electron volts per ion was somewhat high and might typically reduce to the range of 400 to 500 with further use. In addition, another factor strongly dictates the choice of ion-chamber discharge voltage. A minimum discharge voltage should be used to minimize cathode erosion and therefore maximize lifetime.

Accelerator impingement. - Figure 8 shows a typical curve of accelerator impingement versus net accelerating voltage for the flight thrustor. Conditions of the test were: beam current, 0.030 ampere; neutral propellant flow, 0.040 equivalent ampere; discharge voltage, 35 volts; and average screen-to-accelerator-grid gap, 0.156 centimeter. The net accelerating voltage was varied from 7000 to 1500 volts. The ratio of net to total accelerating voltage was maintained constant at 0.8. Below 2150 volts the impingement current rapidly increased with decreasing voltage. The low value of impingement current over the voltage range of 2150 to 5000 volts should permit operation within this range with long accelerator grid lifetimes.

Neutralizer. - From previous experience, it was possible to position the neutralizer within 0.2 centimeter of the best radial location. The final position was a compromise between deep immersion in the beam where there was high erosion of both the boron nitride shield and neutralizer cathode and a withdrawn position where space-charge-limited neutralizer currents cause high coupling voltages. Even a heavy 1.5-centimeter-thick boron-nitride shield cannot withstant the impingement of the normal beam current densities for more than several hundred hours and must be located at the low ion density fringes of the beam.

For all the neutralizer tests the thrustor was operated at net and total accelerating voltages of 4000 and 5000 volts, respectively. Placing the neutralizer tangentially at the nominal (2.5-cm radius) edge of the beam resulted in a coupling voltage of about 200 volts between the beam potential (floating target) and the neutralizer cathode (ground). Moving the cathode radially 1 centimeter outward raised the target voltage to approximately 1000 volts. This voltage was great enough to cause anomalous behavior of the floating target and the assumption of equal beam and target potential was no longer valid. Moving the neutralizer to 0.1 centimeter less than the nominal beam radius resulted in a lowering of the coupling voltage from 200 volts to the values shown in figure 9 (about 35 V). The circumferential position becomes important also, as a critical distance of 0.1 centimeter was small compared with hole spacings of 0.6 centimeter.

At the final neutralizer position the beam current from the thrustor was varied, and the differences in neutralizer emission and coupling voltage were noted. The heating power of the neutralizer cathode was held constant and somewhat higher than emission limited temperatures were maintained. (The neutralizer cathode heating, even after correcting for end conduction losses, generally required 20 percent more power per unit area than did the chamber cathode.) The emission of the neutralizer cathode was directly proportional to the beam current and at a value slightly less than the beam current (fig. 9(a)). The difference between neutralizer and ion beam currents probably represents electrons that were drawn into the beam from other sources within the vacuum tank. The coupling voltage increased with increasing beam current as shown in figure 9(b) until it leveled off near 35 volts and 0.040 ampere. The scatter in the coupling voltage data was typical and might have been caused by local differences in the cathode activation. Previously inactive cathode areas may become activated by ion bombardment, and as space-charge-limited flow propably exists, a larger emitting area would require less coupling voltage.

Because of the sensitivity of the neutralizer cathode and shield position, most tests were considered successful if the coupling voltage were in the range of 50 to 200 volts. The unusually low voltages obtained in figure 9 may be indicative of too little shielding or too much immersion of the cathode in the beam to realize a long neutralizer lifetime.

Propellant flow calibration. - The porous tungsten plug that gave the proper flow rate at the desired temperature level had (according to the vendor) capillaries with a 3.8-micron pore radius and contained 1.6×10^6 pores per square centimeter. The porous tungsten plug tip (shown schematically in fig. 10) was flow calibrated by operating it at a constant temperature and weighing the weight loss from the reservoir. Mercury flow rates as a function of temperature for two tips made from identical porous tungsten are also shown in figure 10. Flow rates for a given plug were reporducible to ±3 percent. After 100 hours of operation, the flow rate was again checked and found to be essentially unchanged. There was negligible heat conduction back to the reservoir, 15 centimeters away. In a seqarate test, the liquid head pressure was increased to 2 atmospheres, and no liquid mercury was forced through the porous plug. Using 8 watts, the plug temperature increased from 400° to 580° K in 30 seconds, and cooled down to 420° K, 45 seconds after the power was turned off. (The temperature cycling range corresponded to two orders of magnitude change in flow rate). The temperature response of the feed tip was judged to be somewhat slow for an altitude control thrustor and would probably need improving before use in this application.

Magnetic fields. - The measured magnetic field strengths for the electromagnetic flight-type thrustor and permanent magnet flight-type thrustor are compared in figure 11. The electromagnetic field was produced by a single large solenoidal coil. The tapered field (lower field in the downstream direction) was found in previous tests (ref. 8) to aid the discharge chamber performance. As a result of weight and size restrictions, the permanent magnet field shown in figure 11(b) contained a hump at an axial position of about 2.5 centimeters from the distributor, rather than the continuously diverging field produced by the solenoid. The permanent magnet field was also measured 1.3 centimeters from the centerline to give an indication of the radial gradient. It was found that the field strength at the 1.3-centimeter radius increased about 5 percent over the centerline values for most of the axial length. Greater increases near the distributor end were probably due to shape of the distributor pole piece (shown sketched in fig. 3).

Figure 12 compares the performance with the two thrustors in terms of the dependence of discharge power per beam ion on the propellant utilization. The discharge power gradually decreases at lower utilizations and rapidly increases at propellant utilization efficiencies higher than 60 or 70 percent. The shape of the field of the permanent magnet thrustor apparently was responsible for the decrease in the maximum propellant utilization, but below propellant utilization efficiencies of 50 percent there was no difference in the discharge losses per beam ion. Some of the difference in the maximum propellant utilization could also have been the result of a variable activation state of the cathode from one test to another. Other tests similar to those shown in figure 12 resulted, after longer running times, in a discharge power per beam ion of 400 to 500 electron volts per ion at a propellant utilization efficiency of 50 percent.

Thrustor Performance and Endurance Tests

Variable geometry thrustor. - A long duration run was conducted to endurance test the oxide-coated brush cathode in a thrustor. A 5-centimeter-diameter thrustor of the type used in reference 6 was utilized for the test. The thrustor was modified by substituting a radial flow distributor for the uniform-flow type. The brush cathode and coating were identical to those later used in the flight-type thrustor. The test proceeded for 1553 hours during which time the thrustor was deliberately shut down and restarted 54 times without removal from the vacuum chamber. The average test values are listed in the last column of table I.

The thrustor ion beam was maintained between 0.015 and 0.020 ampere by adjusting the cathode heating power to give an emission of 0.2 to 0.4 ampere. This mean value of the discharge power per beam ion steadily dropped from an initial value of 600 to a final value of 400 electron volts per ion. The discharge voltage was held constant at 35 volts. This potential was a compromise between lower cathode sputtering rates at lower voltages and a lower discharge chamber loss per beam ion at somewhat higher voltages. The cathode heating power slowly rose from about 15 to 22 watts at 1500 hours. (Details of this heating power curve may be found in ref. 3.) The magnetic coil power loss (27 W) was considered unduly high because of an inefficient design. An effective value of 8 watts was therefore used when computing the power-to-thrust ratio value of 256 watts per millipound listed in table I. Also, a value of 8.7 watts was added to the total power to represent the feed system power. (The actual feed system for this endurance test was a steam-heated mercury vaporizer with an orifice plug.)

At 1551 hours there was a large pressure excursion in the vacuum facility to the 10^{-4} torr range. Although emission was reestablished after this excursion, one of the two cathode heater wires was broken, and the other failed after 2 hours of operation. There was little erosion of the heater wires, and the failure was in a portion well protected from direct ion bombardment. Microphotographs of wire cross section indicated a large amount of a second phase, which may have been tantalum nitride or oxide. This would indicate probable failure by gas embrittlement. At least half of the oxide coating still remained on the cathode.

Flight thrustors. - An initial test was performed on the complete flight-type thrustor, which included a porous plug feed system, electromagnet coil, and neutralizer. The average values of this test are also listed in table I. Specific values of beam current, discharge voltage, cathode heating power, and magnetic field were chosen to optimize the propellant utilization, discharge power losses, and cathode lifetime. The resulting power-to-thrust ratio was 247 watts per millipound at a thrust of 0.65 millipound. This power to thrust ratio was somewhat greater than normal because the discharge power was higher due to operation at a higher propellant utilization. Also, the accelerator impingement current (for unknown reasons) was six times its usual value. The neutralizer coupling voltage was also somewhat high at 130 to 200 volts and the net accelerating voltage of the ion beam was reduced from 4000 to 3800 volts when the beam thrust was calculated.

The flight thrustor was next modified to incorporate a permanent magnetic field. The permanent magnetic field thrustor was tested for 13 hours at essentially constant conditions to determine its steady state operating performance. The results of this test are listed in table I. A beam current of 0.0225 ampere was produced with 144 watts to give a power-to-thrust ratio of 222 watts per millipound. The discharge losses and accelerator impingement were normal and much lower than the electromagnet flight thrustor test. Values were estimated for both the neutralizer cathode heating power and propellant feed power, although no correction for neutralizer coupling voltage was made to the calculated thrust. No neutralizer was used, and a steam-heated vaporizer replaced the electrically heated porous plug.

Cathode heat cycling. - For a thrustor to perform an attitude control mission it must be able to be cycled on and off a great many times. This imposes the stress of thermal cycling and the greatest stress will be on that component that has the largest temperature variation, namely, the cathode. Therefore, a series of tests was undertaken to thermally cycle the cathode only. An oxide-coated tantalum brush identical to that used in the flight thrustor was mounted from a pair of copper supports and heated in a bell jar. After an initial period of 1/2 hour to decompose the alkaline earth carbonates, the cathode heating voltage was snapped on and off in a cyclic manner allowing 15 seconds for heating and 15 seconds for cooling. The cathode reached an equilibrium temperature of 1250° K and 17 watts heating power in 10 seconds. During cooling, the temperature dropped below 900° K (limit of the pyrometer) in 5 seconds. The bell jar pressure was normally in the mid-10^{-7} torr range.

The first three tests failed after 5000 to 15,000 cycles. A fourth test, consisting of a bare brush with no oxide coating, was made to separate the effects of any chemical reaction or physical gas absorbtion between the oxide and the tantalum brush from the effects of thermal fatigue.. The bare brush was cycled 84,000 times at which point the test was stopped because it exceeded the 50,000 cycle requirement of the estimated mission. The core wires were brittle, however, and the brush borke apart when it was removed from the holder.

A fifth test, in which a different solvent was used to clean the brush before coating, was attempted with the identical type of brush cathode. This test, similarly run to the first three tests, lasted for 418,000 cycles before the core wires of the brush broke. Loss of oxide from brush, either by evaporation or by spalling away, was negligible.

The key to this improved cathode lifetime was probably not in the solvent used to clean the brush but in a thermal gradient that existed in the cathode. To reach a surface temperature of 1250° K, the interior core temperature must be about 1600° to 1700° K. At 1700° K, the reaction rate between barium oxide and tantalum (ref. 9) becomes (theoretically) significantly high. Chemical reaction rates are very sensitive to temperature, and perhaps local variations in the brush oroxide coating could cause specific brushes to operate either above or below a critical reaction temperature. Nevertheless, the demonstration of 418,000 cycles proves that under the correct conditions a brush cathode should conservatively be able to meet a mission requirement of 50,000 thermal cycles.

CONCLUDING REMARKS

A lower limit for the thrust of an electron-bombardment ion thrustor appears to be about 0.1 to 0.3 millipound for satisfactory performance in a satellite control system. If the chamber diameter is reduced to 2.5 centimeters or lower, wall recombination losses result in a discharge power that becomes prohibitively high. To avoid wall recombinations, the thrustor may be made larger in diameter, but keeping the thrust level constant requires a reduction of the beam current and, hence, plasma density. At these lower densities, the propellant utilization was sharply reduced. A low propellant flow of 0.020 ampere of neutrals in a 5-centimeter-diameter thrustor (corresponding to a thrust of 0.25 mlb) caused severe difficulty in maintaining a discharge and extracting a beam of more than 50-percent propellant utilization efficiency. A good compromise for a mercury electron-bombardment thrustor of 0.5-millipound thrust was a chamber 5 centimeters in diameter and a propellant utilization of 50 to 60 percent. The low propellant utilization is justifible for the intended mission of satellite control.

The permanent magnet thrustor system, consisting of a propellant tank and vaporizer, thrustor, and neutralizer, is considered to be adaqtable to launch environment and mission requirements. It has light weight and is efficient for the low thrust level. Furthermore, the cathode has demonstrated (in a separate thrustor) substantial lifetime in vacuum tank tests. The best power-to-thrust ratio achieved with the complete thrustor was 222 watts per millipound at 0.65-millipound thrust. If other thrustor missions can tolerate or require higher thrust, the same 5-centimeter-diameter flight-type thrustor could be used at a higher efficiency or a lower ratio of power to thrust than obtained herein. Data of reference 2 indicate, in fact, that a 5-centimeter-diameter thrustor operates efficiently up to 4 millipounds of thrust. Based on these data, a 5-centimeter-diameter thrustor, operating at a thrust of 3 millipounds and a specific impulse of 4000 to 5000 seconds, would produce a power-to-thrust ratio of about 200 watts per millipound. (This includes a power loss of 10 watts for the feed system and 15 watts for neutralization.) The accelerator lifetime, however, would be more critical at this higher thrust level.

APPENDIX A

An endurance run was made with a 10-centimeter-diameter permanent magnet thrustor similar to that described in reference 10. This thrustor was operated at a beam current of 0.13 ampere for the first 768 hours and 0.25 ampere for the last 615 hours. A total of 1383 hours was accumulated on a single molybdenum accelerator grid 0.156 centimeter thick. The hole size and pattern were identical to those reported herein for the flight-type thrustor (0.32-cm-diameter holes on a 0.64-cm equilateral triangular spacing). After 1383 hours the accelerator was reweighed, remeasured for thickness, and rephotographed. The average hole diameter enlargement was 0.02 centimeter while the maximum (near the grid center) was 0.05 centimeter. The thickness of the grid was reduced 0.007 centimeter over the center 5-centimeter diameter and a diminishing amount towards the outer edges of the grid. The calculated weight loss from linear measurements was $5.7 \pm .7$ grams, while the measured weight loss was 5.36 grams.

The total impingement on the accelerator grid was 2.35 ampere-hours at an accelerator grid voltage of -1000 volts, a net accelerating voltage of 4000 volts, and an average propellant utilization efficiency of 80 percent. The erosion from this impingement, if continued linearly, would reduce the web material between holes to zero with a total erosion given by the ratio of web thickness to the maximum wear, 0.05 centimeter, times 2.35 ampere-hours or 14.9 ampere-hours. The lifetime of the accelerator was assumed to be the point at which the web thickness was zero.

Using the technique of reference 11, an erosion of 14.9 ampere-hours, and a lifetime of 13,000 hours, figure 13 was prepared. The thrustor diameter becomes a function of thrust and propellant utilization. The accelerator impingement due to charge exchange was calculated by equation B(16) of reference 11. A direct impingement value (estimated from the measured impingement values of the 1383-hr test minus a calculated charge exchange value) was also added to the charge exchange value. It is interesting to note that for a given thrust, propellant utilization, and erosion rate, the accelerator grid lifetime is proportional to the fourth power of grid diameter (eq. (6), ref. 11).

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11. Kerslake, W. R. "Charge-Exchange Effects on the Accelerator Impingement of an Ekectron-Bombardment Ion Rocket," NASA TN D-1657 (1963).

TABLE I. - PERFORMANCE OF MERCURY ELECTRON-BOMBARDMENT THRUSTORS

	Flight-type thrustors		Thrustor cathode endurance test
	Electro-magnetic	Permanent magnet	
Beam current, A	0.023	0.0225	0.018
Propellant flow rate, equivalent amperes of neutrals	0.035	0.047	0.034
Net accelerating voltage, V	4000	4000	4000
Accelerator voltage, V	-1000	-1000	-1000
Discharge voltage, V	25	30	35
Thrust, mlb	[a]0.65	0.65	0.52
Length of test, hr	95	13	1553
Neutralizer cathode life, hr	58	------	
Beam power, W	92.0	90.0	72.0
Discharge power, W	21.6	12.9	8.0
Discharge cathode power, W	19.0	16.0	15.0
Neutralizer cathode power, W	15.5	[b]15.5	[b]15.5
Magnetic coil power, W	8.0	0	[c]27.0
Propellant feed power, W	8.7	[b]8.7	[b]8.7
Accelerator drain power,	6.0	0.9	1.0
Total power, W	170.8	144.0	[c]128.2
Power to thrust ratio, W/mlb	262	222	247
Specific impulse, sec	[a]4050	3020	3340
Weight of complete thrustor system (less propellant), Kg	5.0	3.0	------

[a] Calculated for a net accelerating voltage of 3800 volts (instead of 4000 V.) to correct for neutralizer coupling.

[b] Neither neutralizer nor electrical feed system was used; estimated values.

[c] Design of electromagnet too large; total power reduced by 19 watts.

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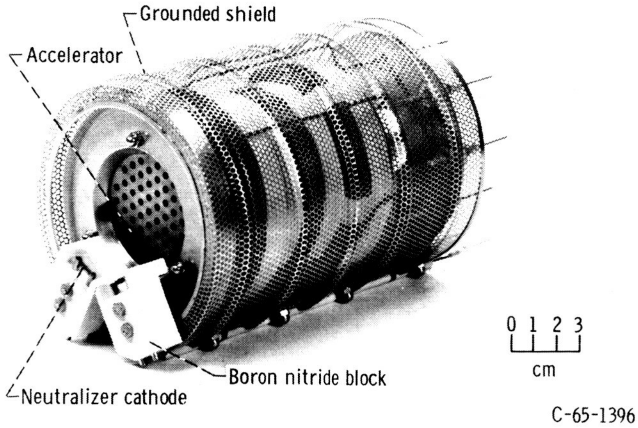
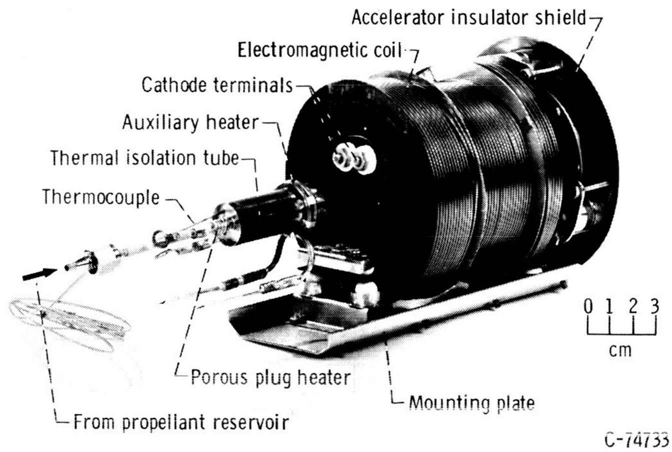
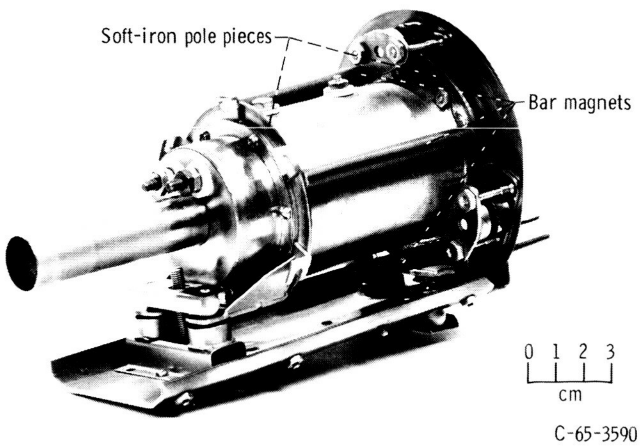


(a) Electromagnetic thrustor, exhaust end.

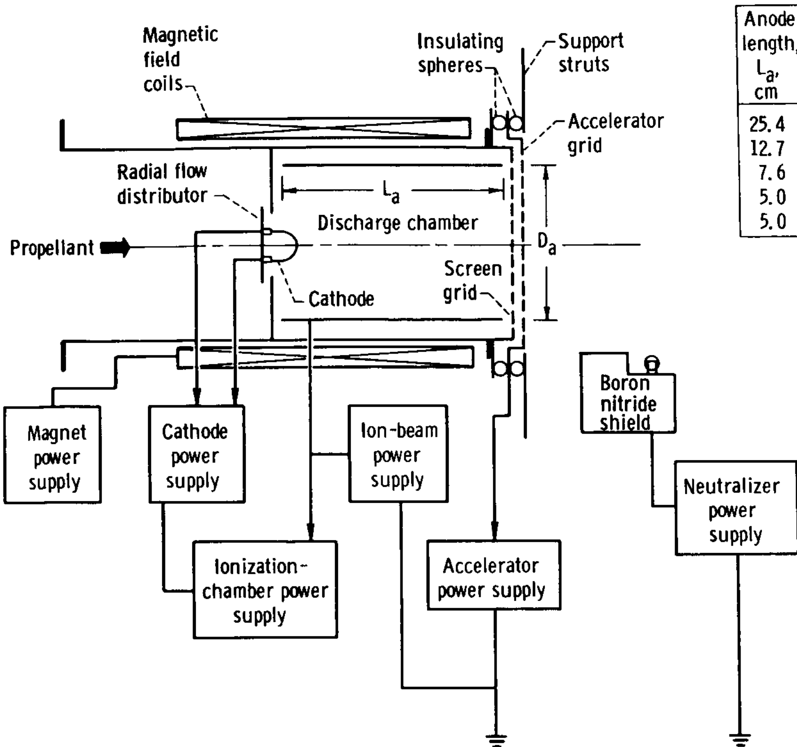


(b) Electromagnetic thrustor, upstream end with outer screen removed.



(c) Permanent magnet thrustor with outer screen removed.

Figure 1. - 5-Centimeter-diameter flight-type thrustors.



Anode length, L_a, cm	Anode diameter, D_a, cm	L_a/D_a
25.4	5.0	5
12.7	5.0	2.5
7.6	5.0	1.5
5.0	5.0	1.0
5.0	2.5	2.0

Figure 2. - Schematic view of variable geometry electron-bombardment thrustor. (Outer screen to reduce stray currents and suppress arcing is not shown.)

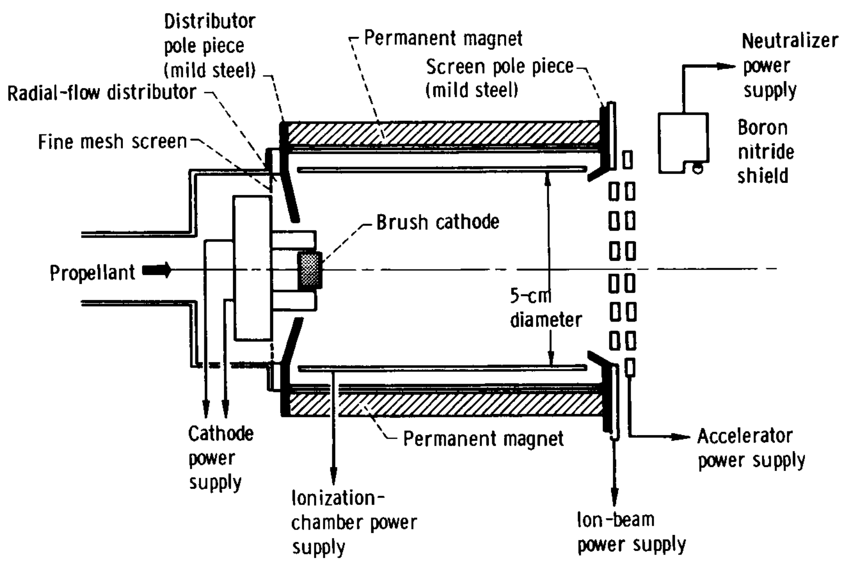


Figure 3. - Permanent magnet flight type thrustor.

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	L/D	Diameter, cm	Magnetic field, G	
			Screen	Distributor
○	2.0	2.5	50	210
□	1.5	5.0	38	120

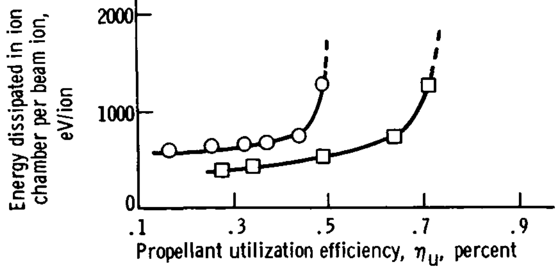

Figure 4. - Performance of two different diameter discharge chambers in variable geometry thrustors. Net accelerating voltage, 4000 volts; accelerator voltage, -1000 volts; discharge voltage 14 to 30 volts; neutral propellant flow, 0.047 ampere.

	Beam current, Ax10^3	L_a/D_a	Discharge voltage, V	Magnetic field, G	
				Screen	Distributor
○	30.1	2.5	22.8	19	63
	30.0	1.5	32.5	19	60
	30.0	1.0	28.6	19	48
□	22.5	5.0	20.8	29	62
	23.0	2.5	21.2	19	63
	22.5	1.5	31.5	19	60
	23.5	1.0	25.5	19	48
◇	16.0	2.5	19.0	19	63
	15.5	1.5	31.8	19	60

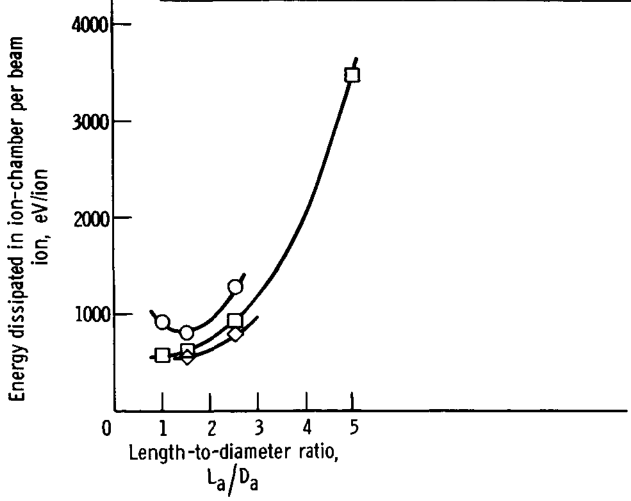

Figure 5. - Discharge chamber length varied in 5-centimeter-diameter variable geometry thrustor. Net accelerating voltage, 4000 volts; accelerator voltage, -1000 volts; neutral propellant flow, 0.047 ampere.

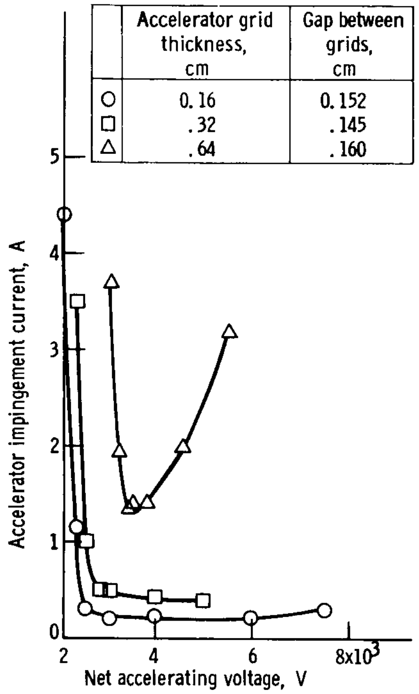

Figure 6. - Accelerator impingement currents versus net accelerating voltage for 5-centimeter-diameter variable geometry thrustor. Screen thickness, 0.16 centimeters; magnetic field at distributor, 100 gauss; at screen, 35 gauss; ratio of net-to-total accelerating voltage, 0.8; ion-beam current, 0.025 ampere.

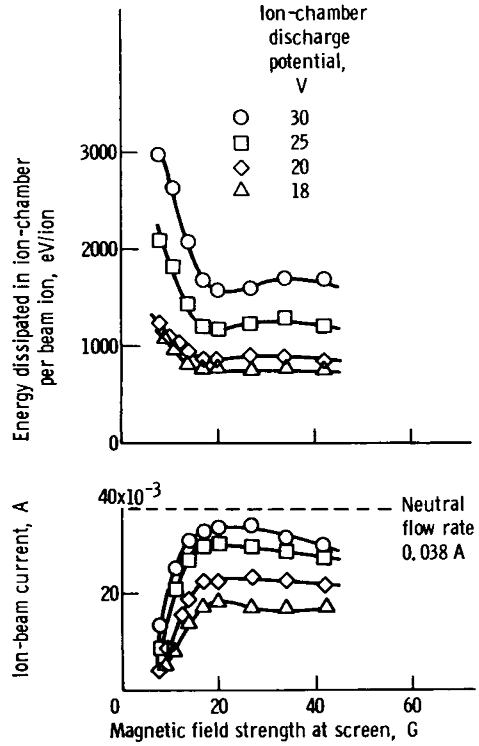

Figure 7. - Discharge chamber performance for the flight-type electromagnetic thrustor. Discharge chamber diameter, 5 centimeters; net accelerating voltage, 4000 volts; accelerator voltage, -1000 volts; cathode heating power, constant at 35 watts.

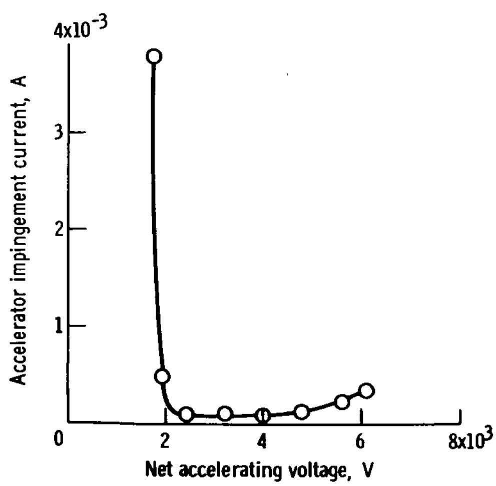

Figure 8. - Net accelerating voltage versus accelerator impingement current for flight-type thrustor. Beam current, 0.030 ampere; discharge voltage, 35 volts; magnetic field, 22 gauss at screen, 60 gauss at distributor; ratio of net-to-total accelerating voltage, 0.8; neutral propellant flow, 0.040 ampere.

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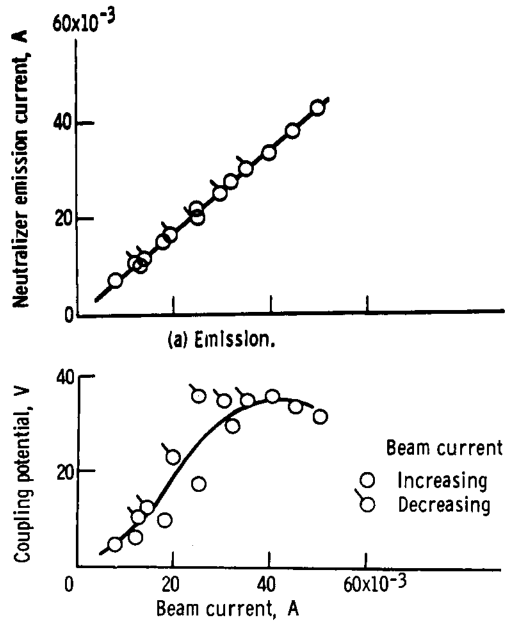


(a) Emission.

(b) Coupling potential.

Figure 9. - Beam current versus coupling potential and neutralizer emission current for flight-type thrustor. Magnetic field strength at screen, 38 gauss; net accelerating voltage, 4000 volts; accelerator voltage, -1000 volts; neutralizer cathode heating power, 15.5 watts; neutral propellant flow, 0.058 ampere.

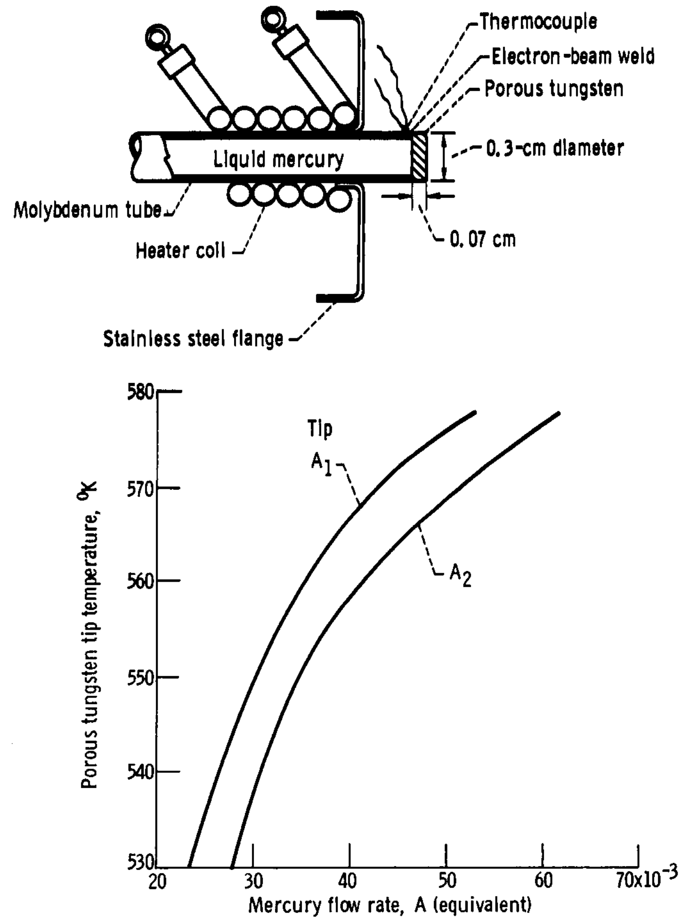
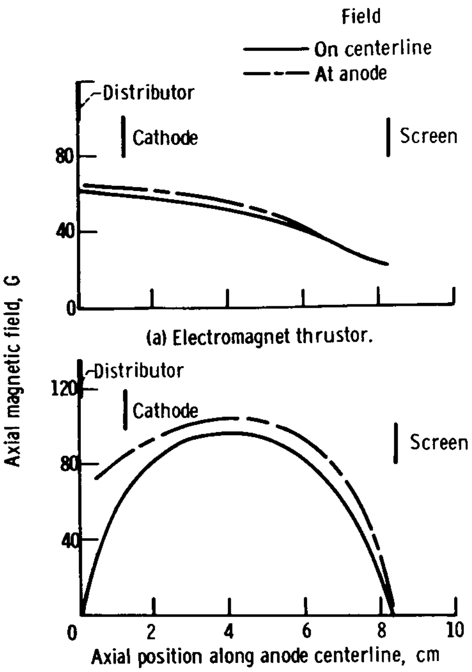


Figure 10. - Calibration of porous tungsten plug tip. Tips 1 and 2 made from the same porous tungsten.



(b) Permanent-magnet thrustor.

Figure 11. - Magnetic field strengths of electromagnet and permanent-magnet thrustors.

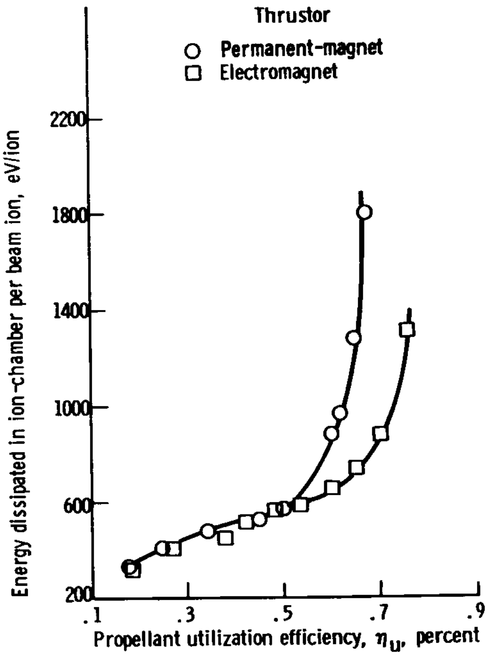


Figure 12. - Performance of flight-type thrustor discharge chambers. Net accelerating voltage, 4000 volts; accelerator voltage, -1000 volts; discharge voltage, 30 volts; neutral propellant flow, 0.050 ampere; magnetic field as shown in figure 1.

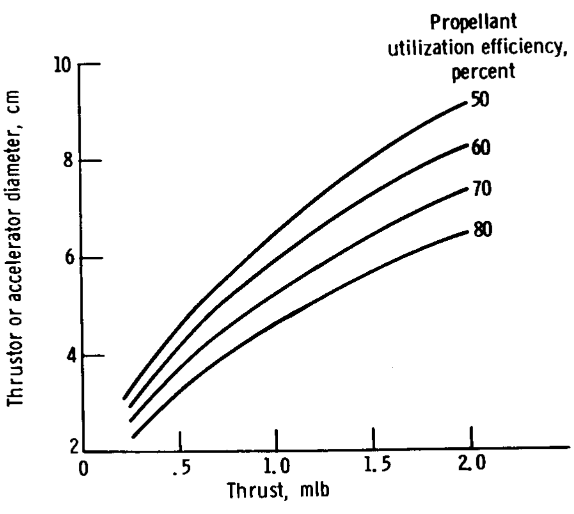


Figure 13. - Calculated thrustor accelerator grid diameter for 13,000-hour lifetime. Impingement included from both direct and charge-exchange ions. Molybdenum accelerator grid had 0.32-centimeter holes on 0.64-centimeter equalateral triangular spacing and was 0.16-centimeter thick.